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(Reno, NV, 6-9 January 2003) **(<u>Deadline: 06 Jan 2003)</u>**

Heat Transfer and Deposition Behavior of Hydrocarbon Rocket Fuels

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Abstract

As the desire to increase the performance of hydrocarbon/liquid oxygen rocket engines naturally leads to increased combustion chamber pressures and higher energy hydrocarbon fuels, the combustion chamber and nozzle heat fluxes also increase. For engines regeneratively cooled with hydrocarbon fuel, this additional thermal stress must be effectively carried by the fuel without degradation of the cooling channel surfaces. A methodology for evaluation of the thermal performance (thermal stability and heat transfer characteristics) of hydrocarbon rocket fuels is suggested. As part of that methodology, an experimental research program to investigate the thermal performance of several new candidate hydrocarbon rocket fuels has been started. The experimental program utilizes a series of test rigs of increasing complexity and fidelity to successively screen identified fuels without the cost and complexity of a full engine system level test. Results of small-scale thermal decomposition experiments utilizing a System for Thermal Decomposition Studies (STDS) test rig provide an initial evaluation of the thermal stability performance of fuels from very small fuel samples. Measurements of heat transfer coefficient and the effect of wall temperature, flow velocity, and wetted-material on deposit formation in heated test channels are obtained from larger rigs, such as the NASA/GRC Heated Tube Facility and the AFRL/PRS High Heat Flux Facility.

Introduction

Regenerative cooling of rocket thrust chambers, defined as cooling by flowing either fuel or oxidizer through passages in the walls of the chamber, was first postulated by Tsiolkovsky in 1903 [Salakhutdinov, 1990]. Thus, 2003 could be described as the "Centennial of Regenerative Cooling". In any case, access to space is dependent upon regenerative cooling of thrust chambers for boosters. This paper focuses on LOX/hydrocarbon thrust engines, as typified in Figure 1. In these chambers, temperatures can exceed 5000 F (3000 K), a temperature well in excess of known material limits. Thus, extended operation (e.g., boosters) requires thrust chamber cooling. Many approaches are available, including regenerative cooling, radiation cooling, film cooling etc. [Sutton et al, 1966]. A major issue in fuel cooling by hydrocarbons is deposit formation ("fouling", "coking"). As regeneratively-cooled engines evolved in the 1950s in the U.S., hydrocarbon propellants also evolved. Initial hydrocarbons used included gasoline, alcohol, and jet fuel (JP-4, JP-5), but these propellants had significant shortcomings. Ultimately, a hydrocarbon fuel was developed

specifically for this application, Rocket Propellant-1 (RP-1), which was a low sulfur, low aromatics kerosene distillate fuel (MIL-P-25576, 1956). This propellant is still in current use. Similar development efforts in the USSR produced a similar, slightly denser propellant known in the U.S. as RG-1 [Mehta et al, 1995] and in Russia as НАФТИЛ (variously "naphthyl" "naphtil" "naftin").

The current issue is how (or if) regenerative cooling limits for hydrocarbons might be extended for high pressure engines. The final developmental F-1 LOX/RP-1 engines for the Saturn V approached 70 atm (1000 psia) in chamber pressure in the late 1960s. The current state-of-the-art in hydrocarbon engines, as exemplified by the RD-180, utilizes chamber pressures in excess of 250 atm (3675 psia) [www.astronautix.com, www. Spaceandtech.com]. This increased pressure helps enable an increased Isp {311 sec (sea level) for the RD-180 vs 265 sec for the F-1}. As discussed below, engine heat flux is roughly proportional to chamber pressure, so this increased engine pressure comes at the expense of increased combustion heat flux delivered to the chamber wall and the fuel. An example of this is shown in Figure 2 [Wagner and Shoji, 1975], where the throat heat flux increases from 10 BTU/in^2-sec (F-1) to about 70 BTU/in^2-sec at 3750 psia. As shown in Figure 3 [Sutton et al, 1966], this increased heat flux may require augmentation of the regenerative cooling by film or transpiration cooling.

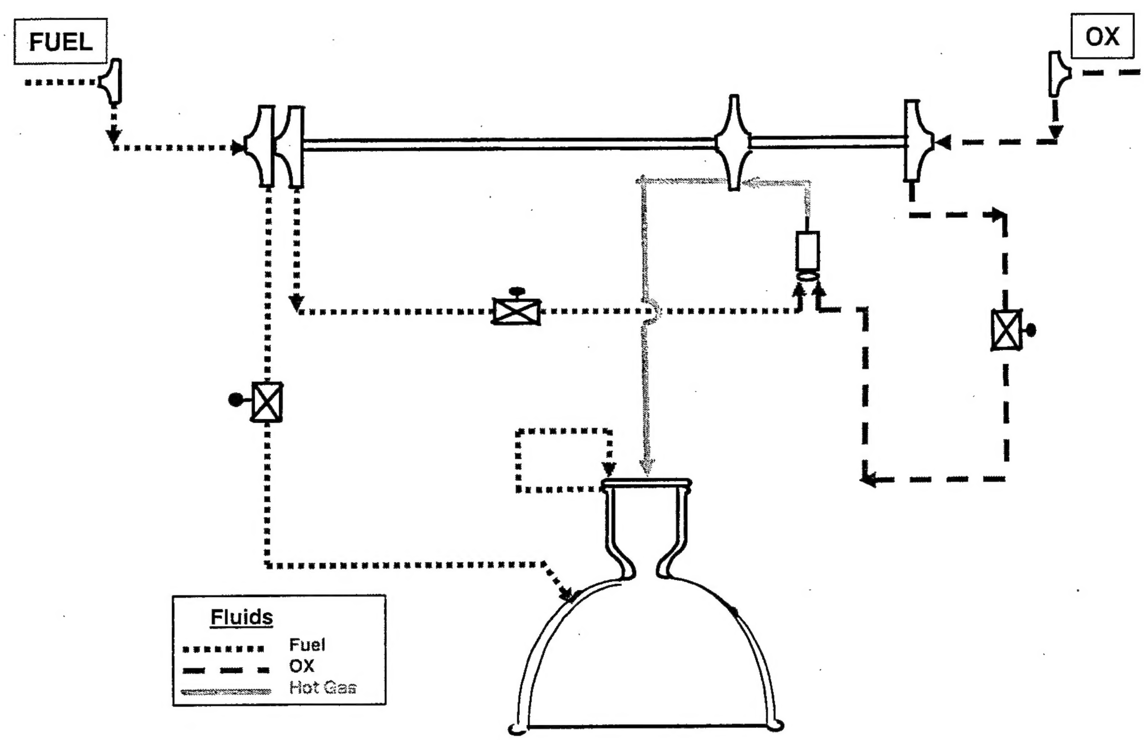

Figure 1. Rocket cycle schematic

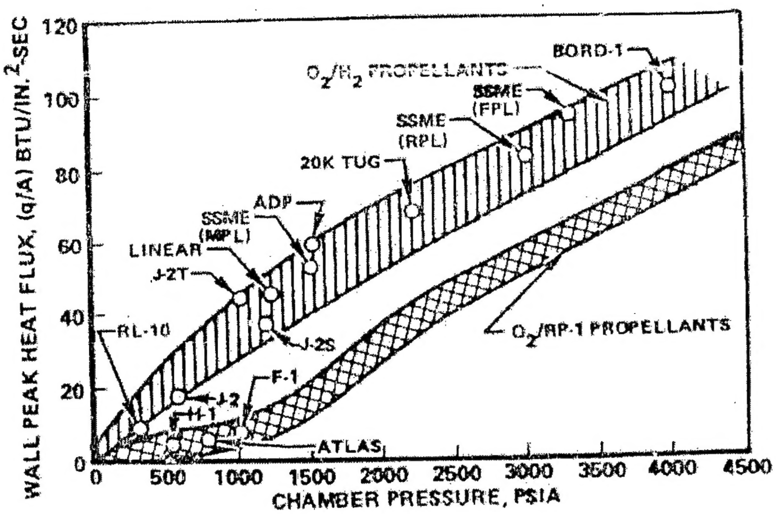

Figure 2. Effect of chamber pressure on nozzle heat flux [Wagner and Shoji, 1975].

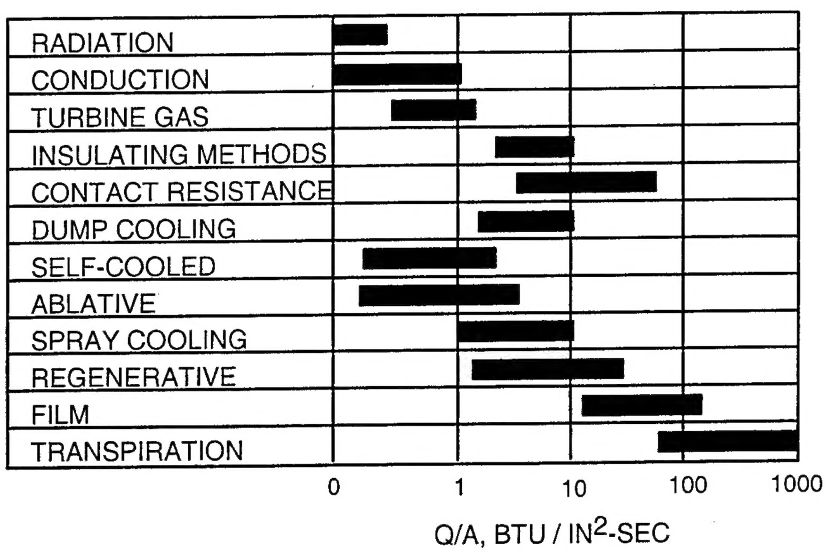

Figure 3. Types of thrust chamber cooling and applicable heat flux range [Sutton et al, 1966].

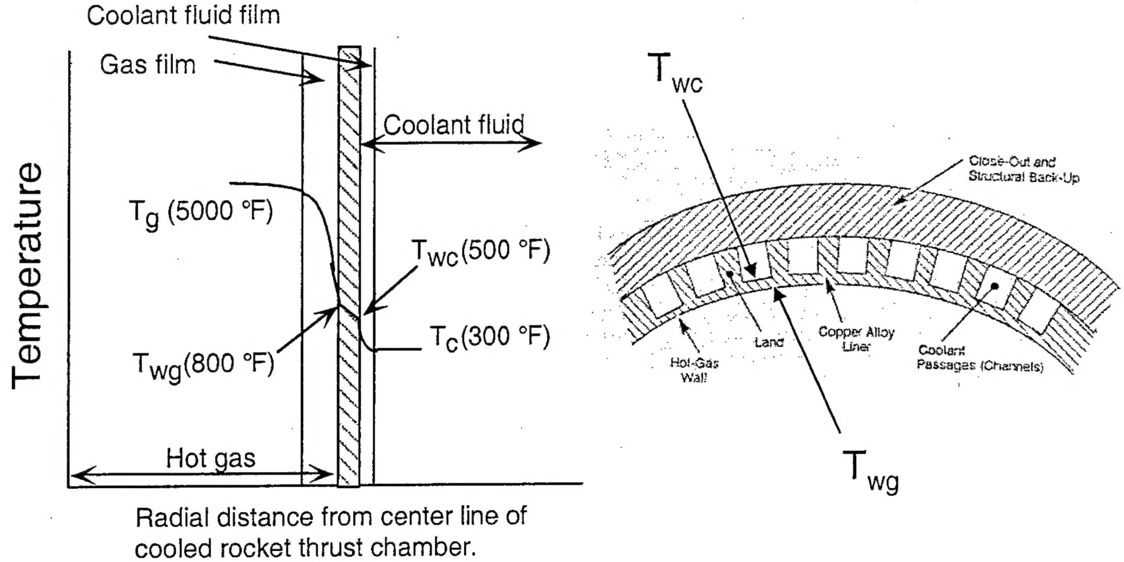

Figure 4. Regenerative cooling outline [Sutton, 1966; Huzel and Huang, 1992].

As shown in Figure 4, regenerative cooling is best expressed as a series of heat transfer processes. Combustion (hot-side) heat flux is calculated as

$$q \sim h_g\,(T_g - T_{wg})$$

where q is the heat flux (e.g., BTU/in2-s, h_g is the combustion side heat transfer coefficient, and the temperatures are as shown in Figure 4. Combustion-side heat transfer is beyond the scope of this paper, but a few comments are relevant.
1) $h_g \sim P^{0.8}$, so the trend of increasing chamber heat pressure increases wall heat flux [e.g., Wagner and Shoji, 1975].
2) the heat flux absorbed by the propellant also increases as $P^{0.8}$, but the propellant mass flow increases as $P^{1.0}$, so the overall temperature rise in the propellant may decrease with pressure. In other words T_{wc} increases with pressure, but T_c at the coolant jacket exit may decrease.
3) Carbon deposition on hot-side chamber walls acts as a thermal resistance and thus reduces heat flux [refs].

On the coolant side, heat flux is a strong function of propellant velocity through the Reynolds number (~V)

$$q = h_c(T_{wc} - T_c)$$

where

$$h_c \sim Re^{0.8}\, Pr^{0.33}(\mu/\mu_w)^{0.14} \text{ (Sieder-Tate)}$$

Fuel heat flux capability is increased by increasing flow velocity and/or careful cooling channel design. High heat fluxes can be absorbed by fluids undergoing nucleate boiling [Bartz, 1958], but current engine designs for hydrocarbons employ pressures well in excess of the critical pressure (~310 psia for RP-1), so this type of heat transfer is not an option. Note that any thermal resistance on the coolant side of the thrust chamber acts to increase T_{wc} for a given heat flux, which can lead to chamber failure ("burn through") if material temperature limits are

exceeded. Carbon deposition is the limiting factor for hydrocarbon propellant heat flux capability [Cook and Quentmeyer, 1980; Wagner and Shoji, 1975]. As shown in Table 1, the excellent insulating properties of "coke" deposits are evident. At the high velocities in regenerative cooling channels (which can exceed 100 ft/sec), carbon deposition can also create increased fuel system pressure drop and result in injection problems.

Table 1. Thermal conductivities of various materials relevant to regenerative cooling. Coke deposit data from [Hazlett, 1991].

Material	k, BTU/hr-ft-F (W/m-K)
Copper	210 (360)
Alumina	3.5 (6)
Superalloy	13 (22.5)
Coke deposit	0.07 (0.12)

Deposition is avoided by keeping the fuel temperature within a "coke limit", which is usually specified as a maximum coolant-side wall temperature (T_{wc} in Figure 4). However, this limiting T_{wc} is not universally agreed-upon or well-characterized in terms of its relationship to coolant velocity, system life, etc. As shown in Table 2, published values vary from 550-850 F.

Table 2. RP-1 Coking Wall Temperature Limit (T_{wc})

Reference	Upper Temperature Limit, °F
Ziebland and Parkinson, 1971	800
Van Huff, 1972	850
Wagner and Shoji, 1975	650 - 700
Wheeler, 1977	600
Cook and Quentmeyer, 1980	600
NASA CR-171712	550

Carbon deposition is often approximated as in Arrhenius form,

$$\text{Deposition rate} \sim \exp(-E/RT_{wc}),$$

yielding plots such as that shown in Figure 5 [Rosenberg et al, 1990]. Such a strong temperature dependence would be difficult to overcome in order to increase regenerative cooling capability of hydrocarbons by increasing T_{wc}. Note also the presence of a velocity-dependence of deposition. As coolant velocity increases at a constant T_{wc}, the boundary layer next to the wall thins and the coolant residence time decreases. Even so, the behavior of kerosene fuels in aircraft heat exchangers and fuel nozzles is much more complex than that shown in Figure 5. Typically, life is again roughly exponentially dependent upon temperature, yet above about 600 F (T_c), the dissolved oxygen in the jet kerosene is consumed and deposition ceases until roughly 900 F. Coking behavior in aircraft occurs under much different conditions than in rockets, as summarized in Table 3.

Several types of hydrocarbons are being examined as possible enhancements/alternatives to RP-1 [Edwards and Meyer, 2002]. Alternative kerosenes tend to focus on higher density, although lower H/C ratios often mitigate any net vehicle benefits [Mills, 2002]. Cryogenic hydrocarbons (e.g., propane) offer increased I_{sp} at the expense of reduced density. High energy (strained ring) hydrocarbons offer high I_{sp} and potentially increased

density, but are relatively uncharacterized and may have toxicity and cost concerns. Note, however, that RP-1 costs for current expendable launch vehicles (Atlas, Delta) are roughly 0.05% of the launch cost.

The rest of this paper describes ongoing efforts to characterize regenerative cooling performance of alternative hydrocarbon propellants. Two major types of devices are used to obtain relevant high-heat-flux conditions: electrically heated tubes and heated copper blocks.

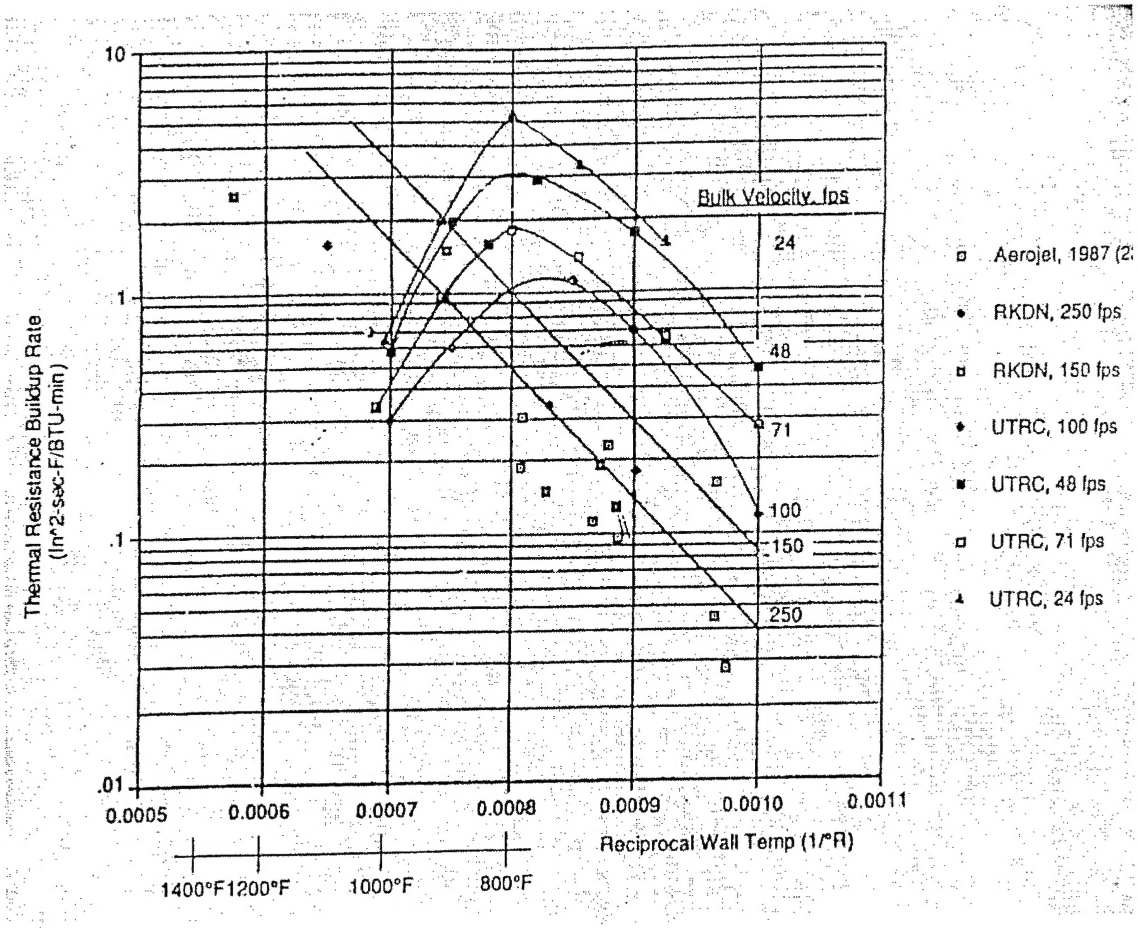

Figure 5 – Typical deposition (thermal resistance) behavior as a function of temperature [Rosenberg et al, 1970].

Table 3 – Comparison of aircraft and rocket cooling conditions

	Aircraft heat exchangers/fuel nozzles	Rocket regen cooling channels
Typical max heat flux, BTU/in^2-sec	1	100
Required lifetime	2000 hrs (min)	300 sec/mission X 50 missions = 4 hours
Materials	superalloys, perhaps coated	Cu alloys
Incompatible materials	Cu!! Cd, Zn, Pb	Cu incompatibility with S in fuel [Rosenberg et al, 1992]
Effect of dissolved oxygen	Oxygen removal below 1 ppm (from typ. 70 ppm) dramatically reduces deposition	Little effect? [Roback et al, 1983]
Maximum deposition rates	JP-8:JP-7~140:1	JP-8:RP-1:JP-7~1:1:1 [Stiegemeier et al, 2002]
Maximum allowable deposit thickness	0.001" (10% flow reduction in typical 0.020" ID passage)	1X10-6" (est) to avoid burnthrough in copper at 100 BTU/in^2-sec
Deposition mechanism	300-600 F – molecular growth through hydroperoxide chain mechanism with acceleration by polar heteroatomic impurities; >900 F – pyrolytic fuel cracking leads to molecular growth through radical chain reactions	??
Specification thermal stability limits?	Yes – ASTM D3241	No

Experimental Measures of Regenerative Cooling Performance (Coking)

The ideal rig test for a propellant's thermal stability (coking tendency) would be:

1) Accurate simulations of the propellant environment in the cooling channels, in terms of heat flux, residence time/velocity, geometry, temperatures, and materials
2) Low cost
3) Low fuel volume required (for developmental propellants that are high cost and/or scarce)
4) Adequately instrumented/analyzed to generate data from which to generate or anchor deposition and heat transfer models

It is quite difficult to fulfill all of these criteria in one test device. High heat flux requires expensive methods to generate the heat and high fuel flow velocities (and thus fuel consumption) to accept the heat. The most test device that generates the most realistic simulation would (of course) be an actual high-pressure, regeneratively-cooled engine, which is certainly not low cost or low fuel consumption. The following sections describe several current experimental efforts to generate fuel stability data under rocket conditions.

Electrically-heated tubes

These are the predominant devices used to obtain high heat flux cooling data, with notable literature references by UTRC [Roback et al, 1983; Giovanetti et al, 1985], Aerojet [Rousar et al, 1984; Rousar et al, 1998], and NASA/GRC [Green et al, 1995; Linne and Munsch, 1995, Stiegemeier et al, 2002]. Chinese work in this area has recently been summarized [Liang et al, 1998]. Currently, only the NASA/GRC rig is active. These types of devices generate heat by passing large amounts of current through a metal tube, which generates heat through electrical resistance. Many propellant studies have been done, notably:

1) Comparison of methane, propane, and RP-1 coolant temperature limits. Propane and RP-1 have roughly comparable limits, while methane exhibits a significantly higher limit.
2) Comparison of the effect of dissolved oxygen
3) Comparison of various kerosene fuels [Stiegemeier et al, 2002]

The NASA Glenn Research Center's Heated Tube Facility utilizes resistively heated test sections to simulate cooling passages of advanced propulsion systems [Green et al, 1995]. The rig has been used in the recent past to investigate a number of heat transfer and fuel thermal stability issues for both liquid rocket engine and high-speed air breathing propulsion systems. The facility has recently been upgraded to better address current programmatic requirements: collection of data with both cryogenic fuels and exploratory HED fuels. The modifications to the facility center around a closely coupled fuel flow system that includes a 17-gallon liquid capacity fuel supply tank rated for 2400 psig. This tank has dual capability for both ambient temperature liquids and cryogenic liquid fuels.

A simplified view of the cryogenic configuration of this new fuel system is shown in Figure 6. The supply tank is filled by passing low-pressure gaseous fuel through a tube coil that is submerged in an open liquid nitrogen bath. As it passes through the coil, the fuel is condensed and thermally conditioned to the approximate temperature required for the test. The supply tank is also insulated and wrapped with a copper coil through which LN_2 flows to trim the temperature of the fuel in the tank. The fuel is pressurized with either gaseous nitrogen or helium brought through a diffuser into the ullage. The fuel is drawn from the tank through a dip tube and passes through a turbine flow meter which is interfaced to the flow control valve through a controller. The test section is contained in a vacuum chamber to reduce heat loss and for safety in the event of a hot fuel leak.

Conduction Heated Rigs

Conduction heated rigs, such as Aerojet's Carbothermal test rig developed under the Hydrocarbon-Fuel/Chamber-Liner Materials Compatibility Program in the late 1980's (NAS 3-25070), offer the advantage of significantly lower electrical power requirements to generate simulative, high heat flux test conditions in cooling channels. Figure 7 shows schematically an example of the device. Unfortunately, the rig no longer exists. AFRL/PRS has undertaken design and construction of a new conduction heated, High Heat Flux Facility, based on the Carbothermal test rig and incorporating extension of maximum heat flux test capability to 100 BTU/in^2-s.

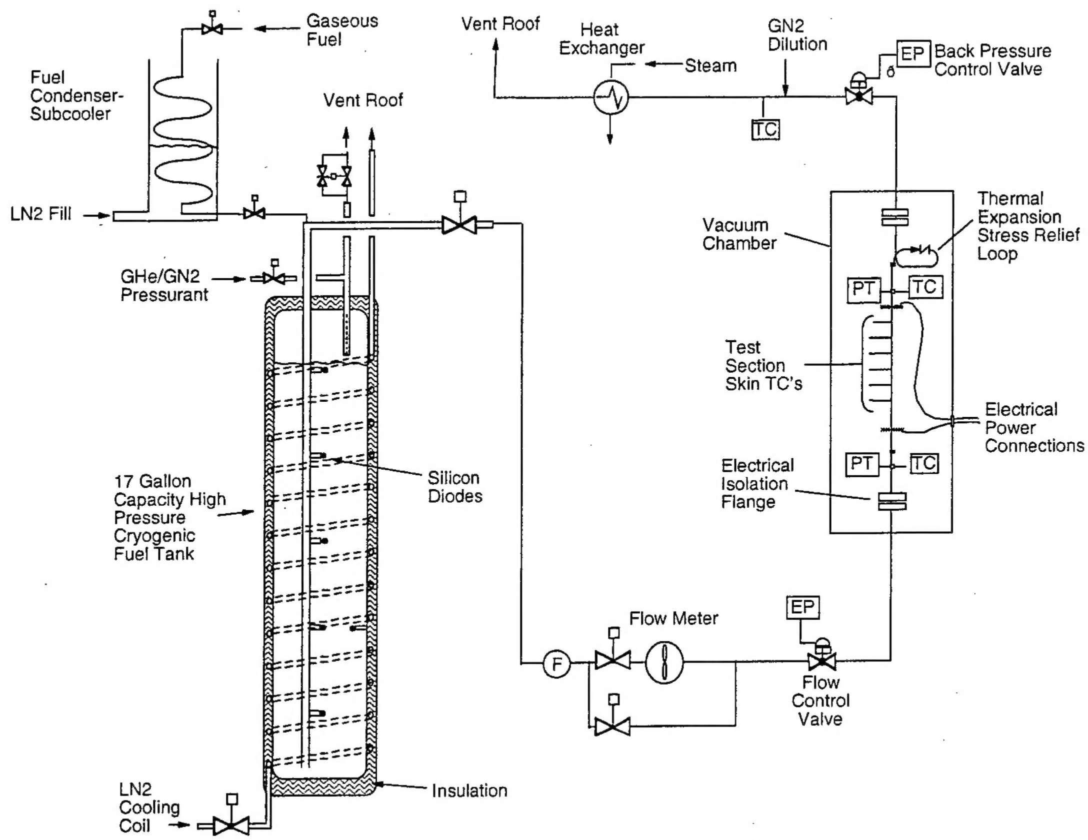

Figure 6. Simplified schematic of the NASA Glenn Research Center Heated Tube Facility's low volume fuel flow loop in cryogenic configuration.

Realistic simulation of engine cooling passages improves with asymmetric heating, three dimensional channel flow in materials and geometries found in high pressure hydrocarbon rocket engines. Pressure, velocity, mass flow rate, bulk temperature, wall temperature, heat flux, axial and temporal heat profiles (spatial and temporal history of flowing fluid entering the test channel), may all be key factors in evaluating a particular fuel's cooling, corrosion, and

thermal stability characteristics. Using the CFD++ code, developed by Metacomp Technologies, Inc, which has the capability of simultaneously handling the conjugate heat transfer problem of 3-d heat conduction in the test device as well as fluid mechanically the flowing channel passage, the detailed behavior of the test channel and fluid can be examined for improved design of experiments. Initial evaluations using the code have reproduced qualitatively heat transfer coefficients within 10% from measured values in experiments reported by Aerojet [Rosenberg et al, 1970]. Ongoing design and analysis activies at AFRL/PRS are addressing improvements in simulative test capability and are expected to yield a device capable of addressing the fundamental mechanisms of deposit formation and material corrosion under high heat fluxes expected in next generation reusable hydrocarbon engines.

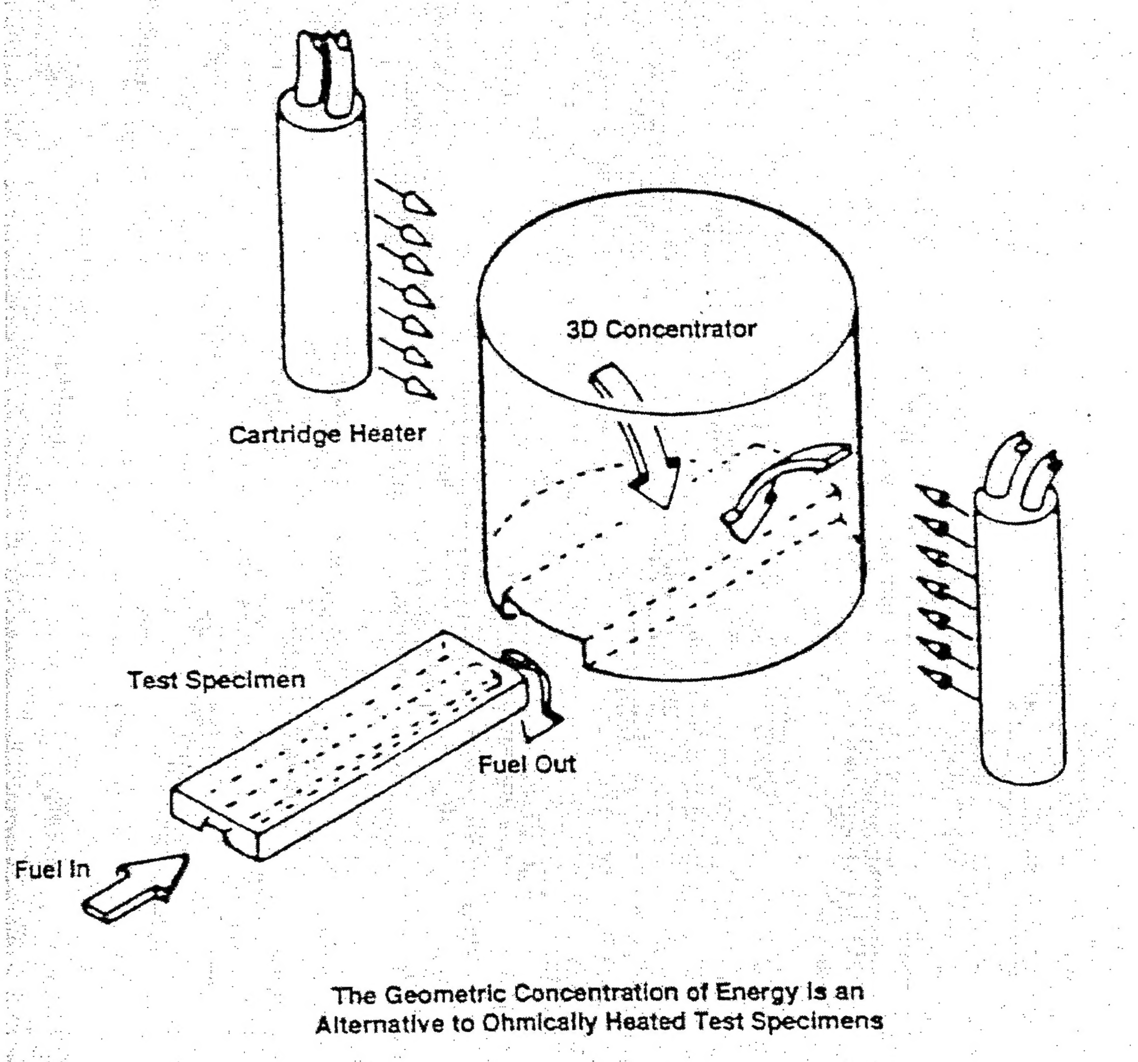

Figure 7 – Schematic of Aerojet's Carbothermal Test Rig (a conduction heated device)

Summary

A methodology of thermal stability analysis incorporating an experimental research program to investigate the thermal performance of several new candidate hydrocarbon rocket fuels has been started. The experimental program utilizes a series of test rigs of increasing complexity and fidelity to successively screen identified fuels without the cost and complexity of a full engine system level test. Results of small-scale thermal decomposition experiments utilizing a System for Thermal Decomposition Studies (STDS) test rig provide an initial evaluation of the thermal stability performance of fuels from very small fuel samples. Measurements of heat transfer coefficient and the effect of wall temperature, flow velocity, and wetted-material on deposit formation in heated test channels are obtained from larger rigs, such as the NASA/GRC Heated Tube Facility and the AFRL/PRS High Heat Flux Facility.

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